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APOLLO LAUNCH VEHICLE DESIGN

by

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FORWARD

This paper is a brief compilation of the many efforts by individuals and organizations in the United States Government and industry who are engaged in the APOLLO program. The contributors are too numerous to name here, but the authors wish to pay tribute to those who made this presentation possible.

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APOLLO LAUNCH VEHICLE DESIGN

By G. P. Pedigo & A. G. Crillion

Project APOLLO, as has been announced, is the manned Lunar exploration program planned by the United States. To accomplish the program objectives a variety of large launch vehicles will be required. Some of the designs of these large vehicles will be summarized in this paper. The designs summarized here constitute the efforts of a large team of Governmental and industrial organizations that are responsible for this important National program. It should be noted that, with the continuing research and development effort that is being applied to these launch vehicles, changes are occurring every day. Some of the launch vehicles described here today will probably have many modifications by the time of final design. Such changes can be expected in a rapidly developing scientific endeavor of this type.

The basic development and operational APOLLO missions are now planned to be conducted on the SATURN C-1 and the SATURN C-5 launch vehicles. The NOVA launch vehicle is being developed to be used as a backup for the SATURN C-5 in the APOLLO program. NASA has selected contractors for each of the major stages of the SATURN C-1 and C-5 launch vehicles. The NOVA launch vehicle is now in the study phase and design contracts are expected to be awarded in the near future. Some of the major mission objectives for these planned vehicles are summarized in Table 1. The design of each vehicle will be optimized for their respective primary mission objectives. The SATURN C-1 primary mission objective is to

place an APOLLO test command module in an Earth orbit. The SATURN C-5 primary mission objective is to accomplish manned Lunar landing and return via rendezvous of two launches. Whereas, for the APOLLO program, the primary mission objective for the NOVA would be to perform the APOLLO Lunar landing and return mission directly with a single launching. Some design features of each vehicle and possible ways that they may be employed to accomplish their mission objectives will now be discussed.

The first orbital flights of the APOLLO test module will be on the SATURN C-1 launch vehicles. The SATURN C-1 APOLLO "A" Space Vehicle, presented in FIG 1, consists of two boost stages, an instrument section and a test module. This vehicle will be approximately 180 feet high. The first stage, called the S-I, has a thrust value of about 1,500,000 lb. The second stage, called the S-IV, is to have a total thrust of about 90,000 lb. Above the S-IV stage is mounted an instrument section and an APOLLO test manned module.

Structurally the vehicle must meet certain design criteria. In general these criteria are dictated by pad, launch and flight loads, each imposing different loading conditions. The structural safety factors to be applied to the computed loading conditions dictated by the largest or governing conditions are shown in Table 2.

For safety and reliability during checkout and launch preparations the C-1 vehicles, as all others, are designed to be free-standing on the launch pad, with the tanks empty or filled, pressurized or unpressurized, 99.9 per cent of the time, during the strongest wind month of the year. This may be defined as being designed to withstand the loads, unsupported

while on the launch pad, imposed by a 99.9 per cent probability-of-occurrence wind profile. The values of such a wind profile at 10 feet are 23 knots steady state with gusts to 32 knots, increasing to 42 knots at 200 feet with gusts to 60 knots.

The S-I stage, shown in FIG 2, has a cluster of nine propellant tanks containing liquid oxygen and the RP-1 fuel (kerosene) used by this stage. The center tank contains liquid oxygen. Circumferentially around this center tank are alternately mounted four additional liquid oxygen tanks and four RP-1 fuel tanks, with a total capacity in excess of 850,000 lb. The S-I stage is powered by eight Rocketdyne H-1 engines. The engine starting sequence is staggered in a 2-2-2-2 arrangement. This is to reduce the dynamic overload of the structure from starting transients while the vehicle is held on the launch pad. A similar cutoff sequence is used if the launch is aborted while the vehicle is on the launch pad. The vehicle is to be held down on the launch pad during first stage thrust build-up until approximately total stage thrust is reached. Since most engine problems usually occur during the thrust build-up phase, holding the vehicle down until near maximum total thrust is reached greatly improves the probability of successful first stage operation. The fins located on the aft section of the S-I stage are necessary for attaining sufficient margin of flight stability. The four 36,000-lb thrust solid propellant separation retro-rockets mounted at the forward end of the stage are used to retard the S-I stage velocity and insure a clean, quick separation of the S-IV stage after first stage powered flight. Innerconnecting liquid level equalizing lines of the S-I stage clustered tank are also depicted in FIG 2.

The S-IV stage, shown in FIG 3, is a cylindrical-type configuration, 220 inches in diameter and powered by six Pratt and Whitney RL10-A-3 engines that use liquid oxygen and liquid hydrogen to deliver a thrust of over 90,000 lb. This stage has a capacity for over 100,000 lb of these high energy cryogenic propellants. Basically, the S-IV stage consists of four major elements: the propellant tanks, the forward interstage structure, the aft interstage structure, and the engine cluster. The liquid hydrogen tank is insulated with the insulating material bonded to the inside surface of the tank walls.

The instrument section, shown in FIG 4, is located on top of the S-IV stage and houses the guidance systems, instrumentation, power supplies, antennas, etc., used for the entire launch vehicle. However, each stage is also equipped with its own instrumentation, power supply, etc., that are in direct service to that stage. A controlled environment for the components is maintained in the X-shaped cylindrical portion of the instrument section. The APOLLO test module is mounted on top of this instrument section.

A typical circular Earth-orbit launch profile for the C-1 APOLLO is shown in FIG 5. This is the type trajectory that is to be used to orbit the manned APOLLO test module around the Earth. The second launch profile shown is a high-speed re-entry trajectory. This trajectory will cause the command module to reach a velocity of approximately 32,000 fps. Entering the atmosphere at this high velocity will approximate the high

temperatures expected in the Lunar mission re-entry phase. This trajectory will enable the heat protection equipment on the command module to be tested and evaluated.

The SATURN C-5 launch vehicle is the second generation of SATURN-class vehicles and is planned to accomplish the APOLLO Lunar missions. The first two boost stages are named the S-IC and the S-II. A third stage, or payload, depending on the particular mission application, is mounted above the second or S-II stage. In the APOLLO program there are different payload concepts being considered for the two missions planned. The first manned APOLLO Lunar mission, which is the circumlunar mission, is presently envisioned to be performed using a two-stage to escape SATURN C-5 vehicle, i.e., the second stage will eject the APOLLO Spacecraft into the circumlunar trajectory. This circumlunar vehicle is depicted in FIG 6.

The second mission planned is a manned Lunar mission. A procedure for this mission, using the SATURN C-5 launch vehicle, may be considered to be via rendezvous of two C-5 vehicle payloads in an Earth orbit. This rendezvous method is necessary because it is presently estimated that, to perform the APOLLO Lunar mission, approximately 150,000 lb of payload must be accelerated to the velocity required to reach the Moon. To do this a total orbital departure weight of about 400,000 lb is needed. The payload weight limit of the SATURN C-5 is approximately 250,000 lb into a low Earth orbit. Consequently, a mission of this type will require two launchings and a mating (or rendezvous) of the orbital payloads to meet the

orbital weight requirements. Two methods for the orbital rendezvous being considered are: a Connecting Mode and a Tanking Mode. Each would involve two sequential launchings.

Two typical launch configurations for the connecting mode are shown in FIG 7. Each vehicle would have the same first two boost stages, i.e., S-IC and S-II. The first launch vehicle payload (on top of the S-II) would be the fueled R-1 stage. This stage would first be injected into a circular orbit by the S-II stage. The second launch vehicle payload would be the APOLLO Spacecraft. In sequel to the first launch vehicle payload, this second launch vehicle payload would be placed in a higher circular orbit. Tracking and status data from each payload are fed into an Earth-based computer and, at a subsequent time of near coplanar orbits and proper phasing, the R-1 stage is commanded to commence the rendezvous maneuvers. A thrust is then applied to the R-1 stage, causing it to ascend in a near elliptical flight path to the higher circular orbit. As the ascending stage nears the Spacecraft, it is monitored by the Spacecraft. The velocity, distance, and attitude data are analyzed and corrective maneuvers are commanded. The rendezvous would be accomplished as schematically shown in FIG 8. At the appropriate time, after rendezvous is completed, the R-1 stage is started, accelerating the Spacecraft to the necessary velocity for accomplishing the mission. When the proper velocity is reached the R-1 stage is then discarded.

The tanking mode vehicles, as in the connecting mode vehicles, would consist of the S-IC and S-II boost stages (FIG 9). The first vehicle launched would have a tanker payload containing the oxidizer for the R-1 stage and is parked in the lower circular orbit. The second launch would then place into a higher circular orbit the Spacecraft plus the R-1 stage

with no oxidizer. The rendezvous maneuvers are performed similarly to the connecting mode and then the liquid oxygen is transferred from the tanker to the R-1. At this point the R-1 stage and Spacecraft would proceed as in the connecting mode. FIG 10 schematically displays the tanking mode rendezvous.

The tanker mode appears to offer a performance advantage over the connecting mode in that most of the orbital rendezvous equipment, such as docking structure, maneuver equipment, etc., can be ejected after the propellant is transferred. This advantage cannot be fully realized in the connecting mode. However, remote transfer of cryogenic propellants, under weightless conditions, presents problems of considerable complexity.

It has been shown that each of the SATURN C-5 launch vehicles has a common first and second stage, S-IC and S-II. The design configurations of the APOLLO Spacecraft and the R-1 stage will depend on the type mission and the flight modes to be employed. There are many concepts as to how the orbital rendezvous or direct ascent technique could be used in the APOLLO Lunar missions. Each concept imposes different design requirements on the launch vehicle. Studies and analyses on these different concepts are continuing and the final design of the launch vehicle will depend on which concept is chosen. In the design studies similar criteria to those mentioned in the SATURN C-1 discussion are imposed. This is in regard to structural, holddown, wind load, and factor of safety requirements. Some aspects of the various launch vehicle stages discussed will now be described.

The S-IC stage, illustrated in FIG 11, will be approximately 140 feet long, 33 feet in diameter, and powered by five F-1 engines using liquid oxygen and RP-1 fuel as propellants to deliver a sea level thrust of

7,500,000 lb. The propellant containers will have a capacity of about 4,600,000 lb.

The basic propellant container design will feature a cylindrical structure that has separate bulkheads for the propellant containers, with the liquid oxygen tank forward and the RP-1 tank aft. The separate bulkhead will increase the stage overall length but is necessary for manufacturing and reliability reasons. Each container will have slosh suppression devices.

A corrugated or honeycomb cylindrical section connected with two ring frames located at each end will form the thrust structure. The upper ring frame will distribute horizontal and launcher loads from four posts. Vertical loads introduced by the engines are distributed through the thrust structure by the four posts. The launcher holddown loads are distributed to the thrust structure by the launcher posts. Vertical loads introduced by the center engine are carried out to the ring frame of the thrust structure by a cross beam system.

The four outboard engines are to be mounted on a diameter of 364 inches. An aerodynamic fairing will be located over each outer engine. A separate liquid oxygen suction line, connecting each engine to the liquid oxygen container, will run through insulated tunnels in the fuel container. Two fuel suction lines connect each engine to the fuel container. Both the liquid oxygen and fuel lines will be routed for the best propellant utilization characteristics. To minimize the amount of suction line ducting and the loads imposed on the engine, pressure balanced zero-fluid displacement flex joints are used. Fuel container pressurization will be accomplished with helium. The gaseous helium is stored in liquid nitrogen jacketed spheres to increase storage quantity. From the spheres it is ducted through engine mounted heat exchangers to raise its temperature

before being introduced into the fuel tank. Gaseous oxygen generated in the engine mounted heat exchangers may be used for pressurizing the liquid oxygen tank. ^{However} this scheme may be replaced by a helium system similar to that used by the fuel tanks, if weight and reliability advantages are realized. Gimbaling is accomplished by double-acting piston gimbal actuators using the RP-1 fuel, bled from the high pressure propellant feed system, as the hydraulic fluid. Engine start sequence is to be 1-2-2 with the center engine first. As in the case of the C-1 first stage, this reduces the dynamic overload of the structure during the start transients. A similar cutoff sequence is used if the launch is aborted while the vehicle is on the launcher. This permits substantial weight savings in the structure. In-flight engine cutoff sequence is presently anticipated to be 1-4; likewise with the center engine first.

The S-II stage, depicted in FIG 12, will be approximately 80 feet long, 33 feet in diameter, and equipped with five J-2 rocket engines that operate on liquid oxygen and liquid hydrogen. Four of the engines will be mounted on a diameter of 210 inches, with the fifth engine mounted in the center of the thrust structure. The propellant containers will be designed for a propellant capacity of about 800,000 lb.

The basic propellant tank structure is of a conventional semimonocoque design with a common, insulated, doubled walled bulkhead separating the liquid oxygen container from the liquid hydrogen container. The liquid

hydrogen is to be located above the liquid oxygen. The common bulkhead will be designed to structurally withstand a liquid oxygen container pressure drop to ambient without failure.

The aft interstage structure will transmit the thrust loads from the S-IC stage to the aft skirt structure; the aft skirt structure will transmit the loads to the S-II stage body; and the forward skirt structure will in turn transmit the thrust loads from the S-II stage body to the stage above. A conical thrust structure will transmit forces from the J-2 engine mount frame to the aft skirt structure during S-II stage powered flight.

Slosh suppressions devices will be provided in each propellant container to control sloshing motion of the propellants during flight. A propellant utilization system will be provided to control the rate of oxidizer consumption in order to maintain the proper fuel-oxidizer mixture ratios.

Control of the S-II stage will be achieved by gimbaling the four outboard engines to a maximum of $\pm 7 \frac{1}{2}$ degrees, including overtravel for snubbing. The center engine will be fixed. All five engines will be aligned parallel to the centerline of the vehicle.

The R-1 stage will vary, depending on the launching mode used. Basically it will be similar to that shown in FIG 13. The connecting mode propellant tank capacity for the R-1 stage will be about 170,000 lb. The R-1 propellant tank for the tanking mode will be larger, requiring a capacity of about 230,000 lb. For either mode the propellant container design will be of cylindrical shape with a diameter of 220 inches. As in the S-II stage, a common, insulated, doubled walled bulkhead will separate the propellant containers. The Liquid hydrogen will be above the liquid oxygen. The R-1 stage is to be designed to meet the requirements for extended stay-time in orbit as dictated by the particular mode of operation that is yet to be selected. Until the particular mode of operation is established, the R-1 stage design cannot be fully defined.

The "APOLLO Spacecraft" referred to in this discussion is considered to consist of three major components.

1. The R-2 which is the Lunar braking and landing stage.
2. The R-3 which is the return stage, i.e., that stage which would take off from the Lunar surface.
3. The APOLLO command module which would house the astronauts and is the portion that returns to the Earth's surface.

A quantitative estimate of the expected loading conditions that would be experienced in flight, with a 99 per cent probability-of-occurrence wind loading, is offered in FIG 14. This demonstrates the loading profiles that may be expected and must be considered in the design of each stage. The vehicle for this case is the second launch fueling mode vehicle which represents the worst case based on its geometry, i.e., length and diameter.

A backup study is now being made on the C-5 APOLLO launch vehicle to consider the possibility of using a nuclear rocket engine stage. The mission potential of a nuclear stage is large; however, the usefulness of a nuclear rocket must await the development of an engine and stage. The purposes of the NERVA and RIFT programs, now underway, are to develop such a system. A discussion of how the C-5/Nuclear Vehicle may be conceived follows.

The C-5/Nuclear launch vehicle could be similar to the one shown in FIG 15. It would have the same first two boost stages as the chemical C-5. The nuclear stage would be the third stage, replacing the R-1, and would have the same diameter as the S-I and S-II stages.

Because of the high or specific impulse of the nuclear stage, a greater payload performance can be achieved. Therefore, the nuclear stage offers the potential of the SATURN C-5 performing the APOLLO Lunar mission without the complication of orbital rendezvous.

Considering a nuclear powered stage with the capability to deliver the APOLLO Spacecraft payload to the Lunar injection velocity, a typical Lunar trajectory could be as shown in FIG 16. After the boost stages have been fired and separated, the nuclear stage engine is operated to go into a parking orbit. Then the engine is stopped. At the proper time (or launch window) the nuclear stage engine is restarted to achieve the necessary velocity. At this point the nuclear stage may be discarded and the APOLLO Spacecraft would proceed as in the C-5 chemical version.

The NOVA launch vehicle is still in the conceptual design phase and much work is yet to be done before a specific description can be given. A typical NOVA launch vehicle, shown in FIG 17, may be envisioned as being

approximately 360 feet long and consisting of three boost stages with the payload, i.e., the APOLLO Spacecraft, on top of the third stage. This launch vehicle would have an initial or first stage thrust in excess of 12,000,000 lb.

The first stage, N-1, would have a tank diameter approaching 50 feet and would be over 115 feet long, with a propellant capacity of over 6,000,000 lb. It would be equipped with eight or more F-1 engines of about 1,500,000 lb thrust, each using liquid oxygen and RP-1 as fuel.

The second stage, N-2, would also be over 115 feet long with a diameter of about 40 feet. The N-2 stage would be equipped with two or more M-1 engines. The total propellant capacity of liquid oxygen and liquid hydrogen would be expected to be over 1,800,000 lb.

The third stage, N-3, would be about 22 feet in diameter, by about 82 feet in length. It would be equipped with one or more J-2 engines, using liquid oxygen and liquid hydrogen as propellant with an expected propellant capacity of about 250,000 lb.

The payload above the third stage would be similar to the APOLLO Spacecraft used in the SATURN C-5. The mission trajectory of the NOVA may be similar to that shown in FIG 17. The N-1 and N-2 stages would first place the N-3 stage with the APOLLO Spacecraft into a parking orbit. The N-3 stage would then be replacing the R-1 stage. Acceleration to the required velocity would be similar to the SATURN C-5 cases discussed.

In this paper various types of vehicles that may be used to accomplish the APOLLO program have been briefly reviewed. For each of these various vehicles the design requirements or criteria used must be oriented to assure that the mission objectives are met. The design of each vehicle must take into consideration human endurance (as applicable) and have sufficient reliability to insure safety and ability to accomplish the mission objectives.

Table

~~FIGURE~~ 1. APOLLO MISSION OBJECTIVES

SATURN C-1

1. Research and development testing of the APOLLO spacecraft systems
2. Orbital flight test of the APOLLO command module
3. Training of APOLLO crews
4. Re-entry testing of the APOLLO command module
5. Reliability testing of APOLLO spacecraft components in a space environment

SATURN C-5

1. Further development of the APOLLO spacecraft system
2. Perform the circumlunar flight of the APOLLO command and service modules, and return to Earth of the command module
3. Perform the manned Lunar landing and return mission, by the orbit ~~rendezvous~~ rendezvous technique

NOVA

1. Serve as a back-up launch vehicle for the SATURN C-5
2. Perform the manned Lunar landing and return mission directly, using only a single launching
3. Perform manned space explorations sequ^el to the APOLLO

Table

~~FIGURE~~ 2. VEHICLE FACTORS OF SAFETY

1. General Structure

Yield factor of safety = $1.10 \times$ design load

Ultimate factor of safety = $1.40 \times$ design load

2. Vehicle Propellant Tanks

Proof pressure = $1.05 \times$ limit pressure

Yield pressure = $1.10 \times$ limit pressure

Burst pressure = $1.40 \times$ limit pressure

3. Hydraulic of Pneumatic Systems

Flexible hose, tubing, and fittings less than 1.5-inch diameter

Proof pressure = $1.50 \times$ limit pressure

Burst pressure = $4.00 \times$ limit pressure

Flexible hose, tubing, and fittings greater than 1.50 inch diameter

Proof pressure = $1.50 \times$ limit pressure

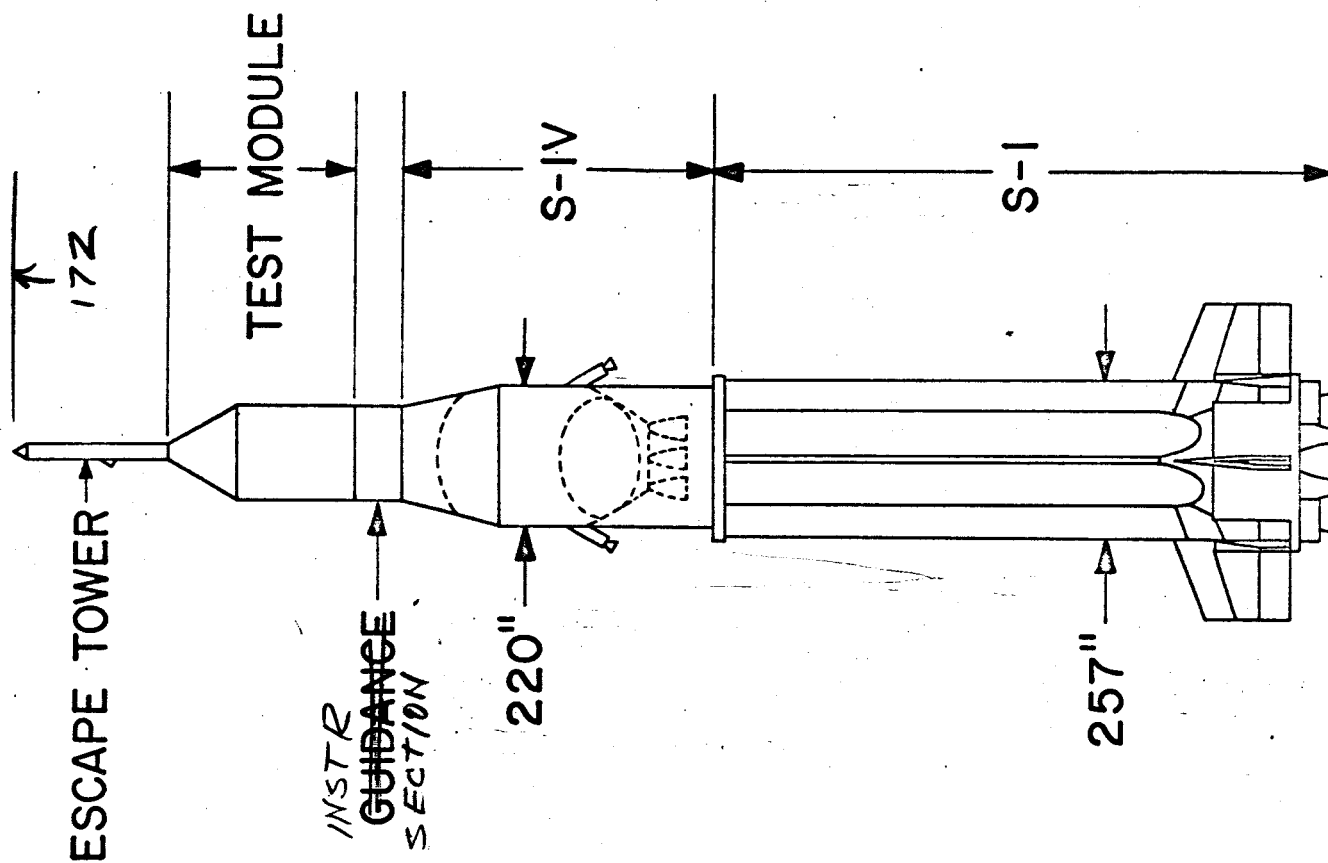
Burst pressure = $2.50 \times$ limit pressure

Air Reservoirs

Proof pressure = $1.50 \times$ limit pressure

Burst pressure = $2.5 \times$ limit pressure

C-I WITH APOLLO TEST MODULE



S-I STAGE

S-I PLUMBING & LOX TANKS

RETROES →

FINIS →

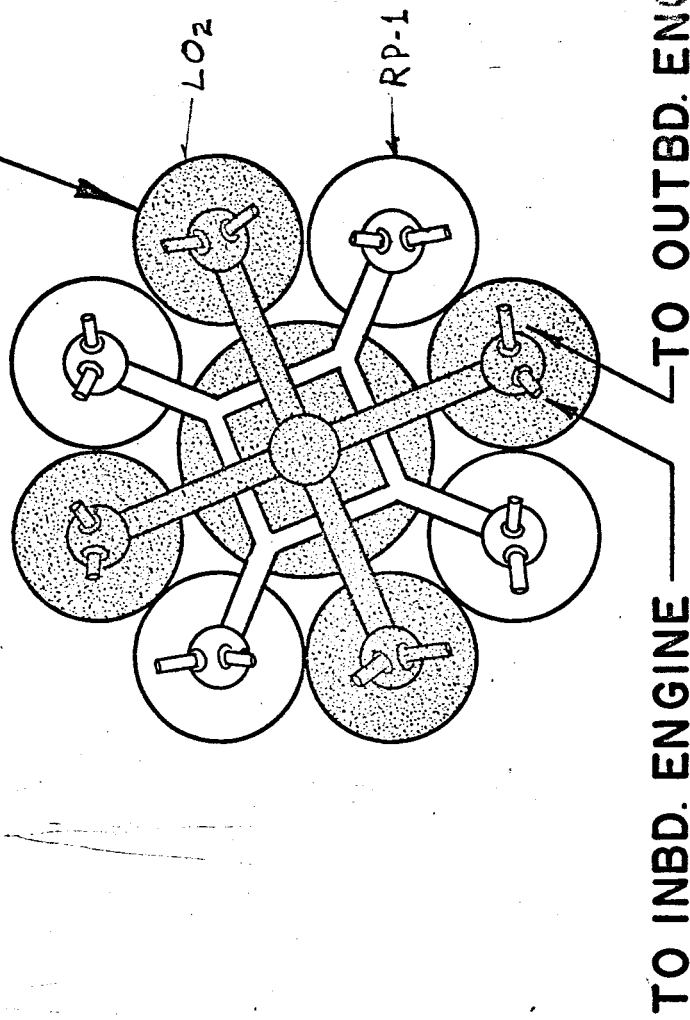
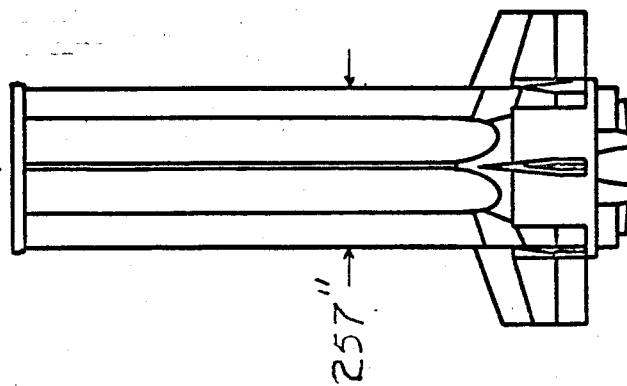
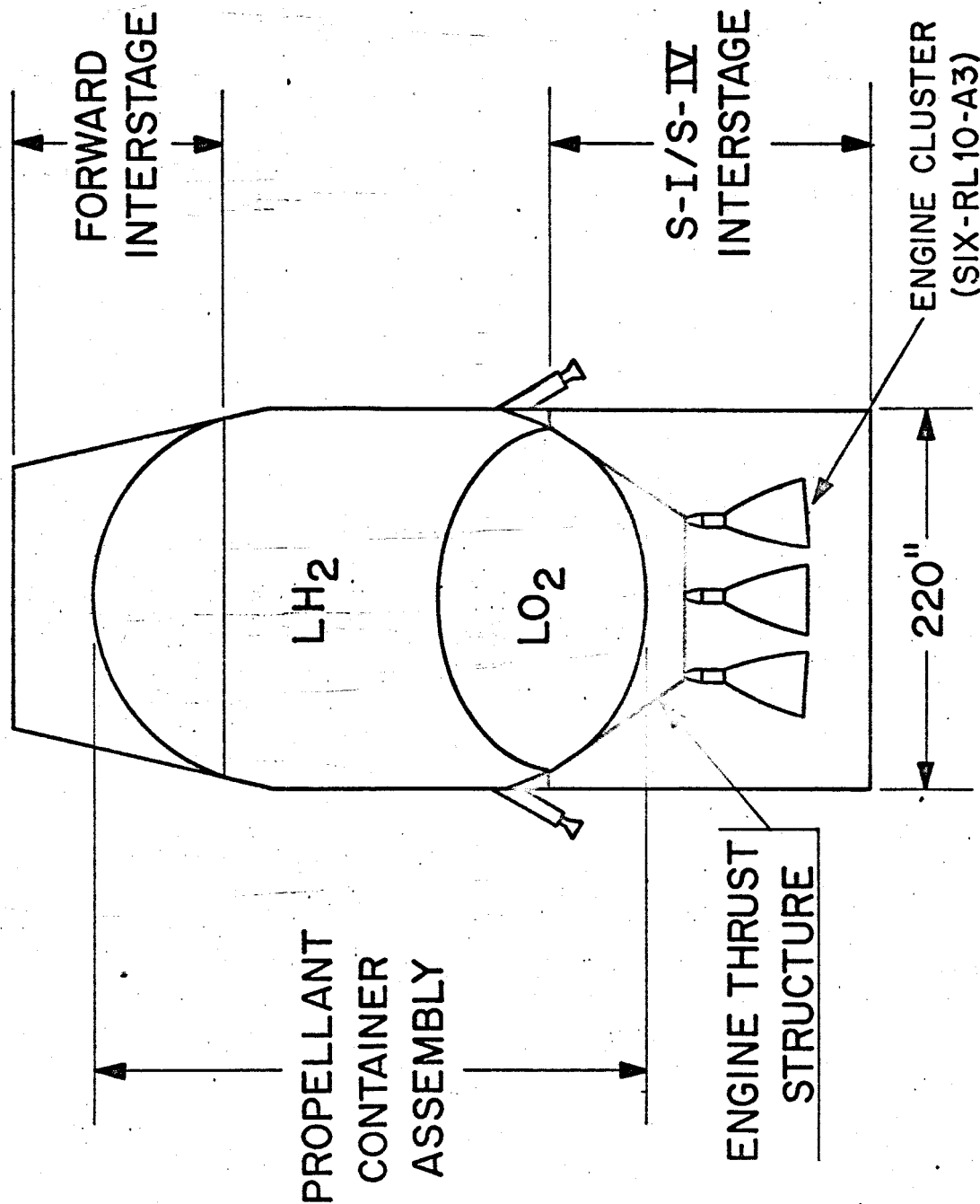


Fig 2

S-IV STAGE



TYPICAL
INSTRUMENT COMPARTMENT
SECTION

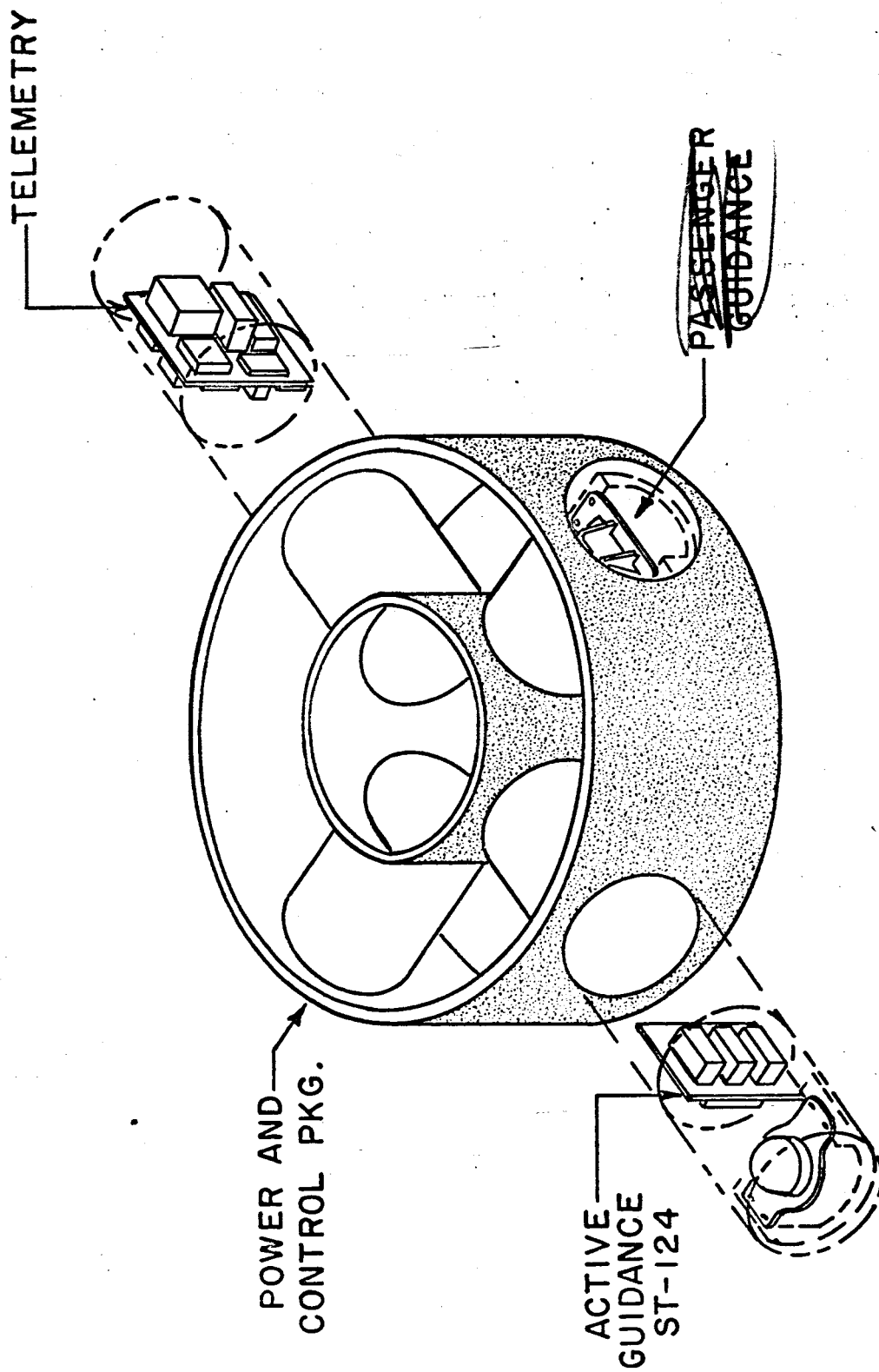
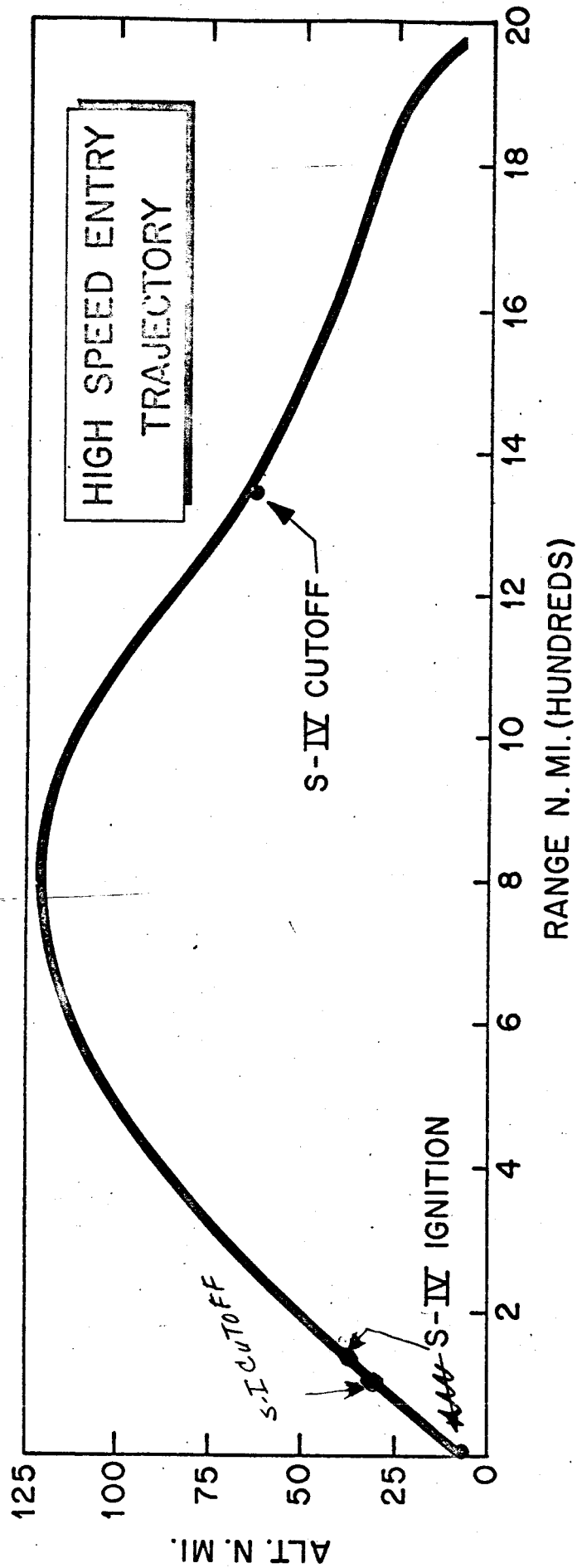
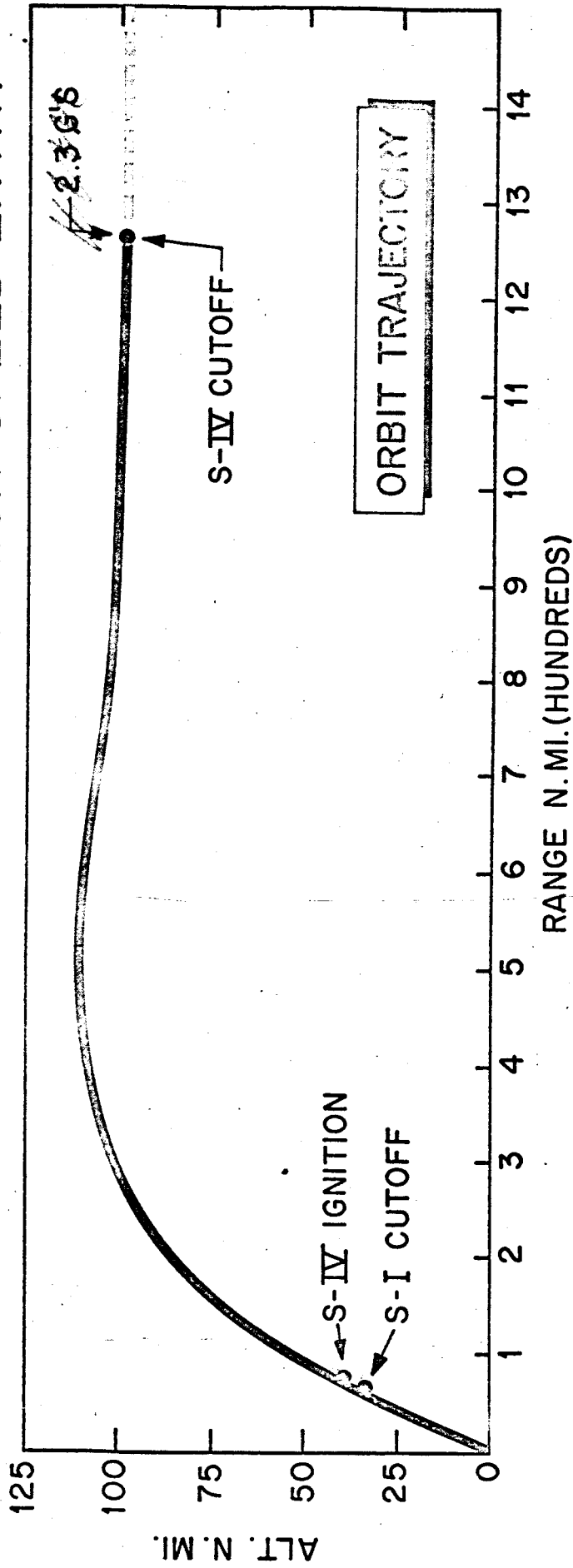


Fig 4

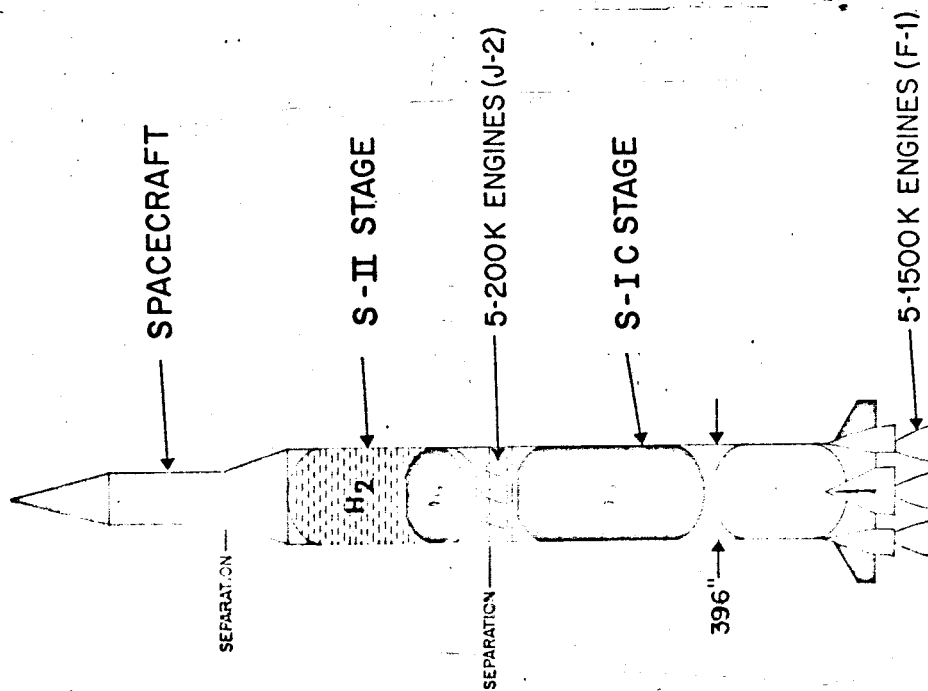
TYPICAL

C-I TRAJECTORIES-ORBIT & HIGH SPEED ENTRY



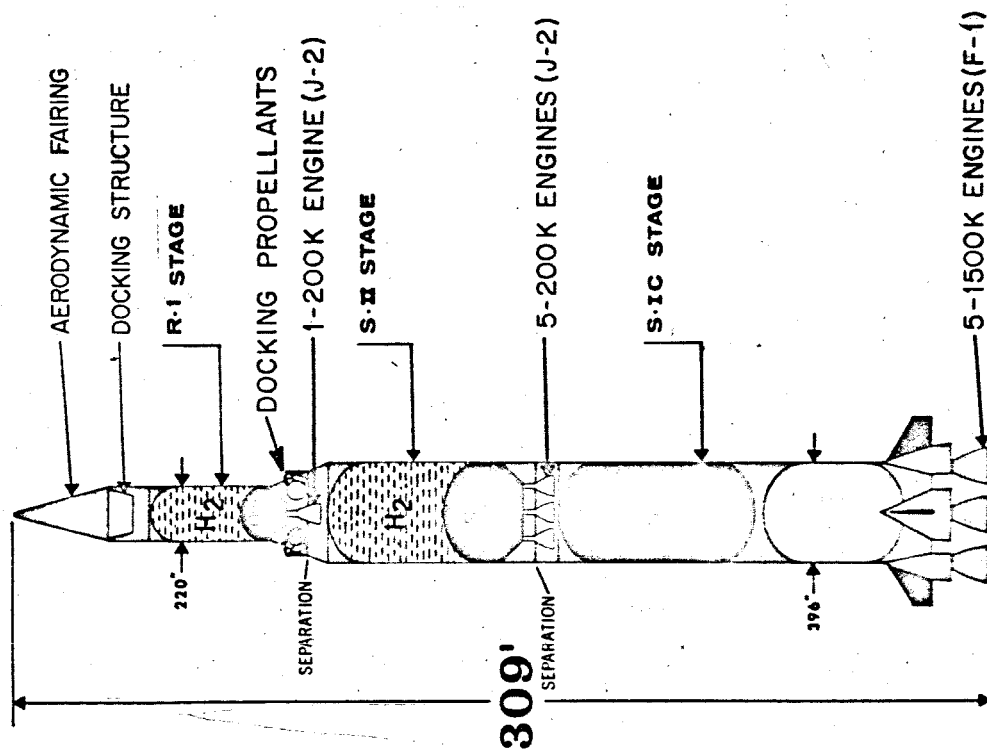
TYPICAL

C-5 CIRCUMLUNAR (2 STAGE VEHICLE)

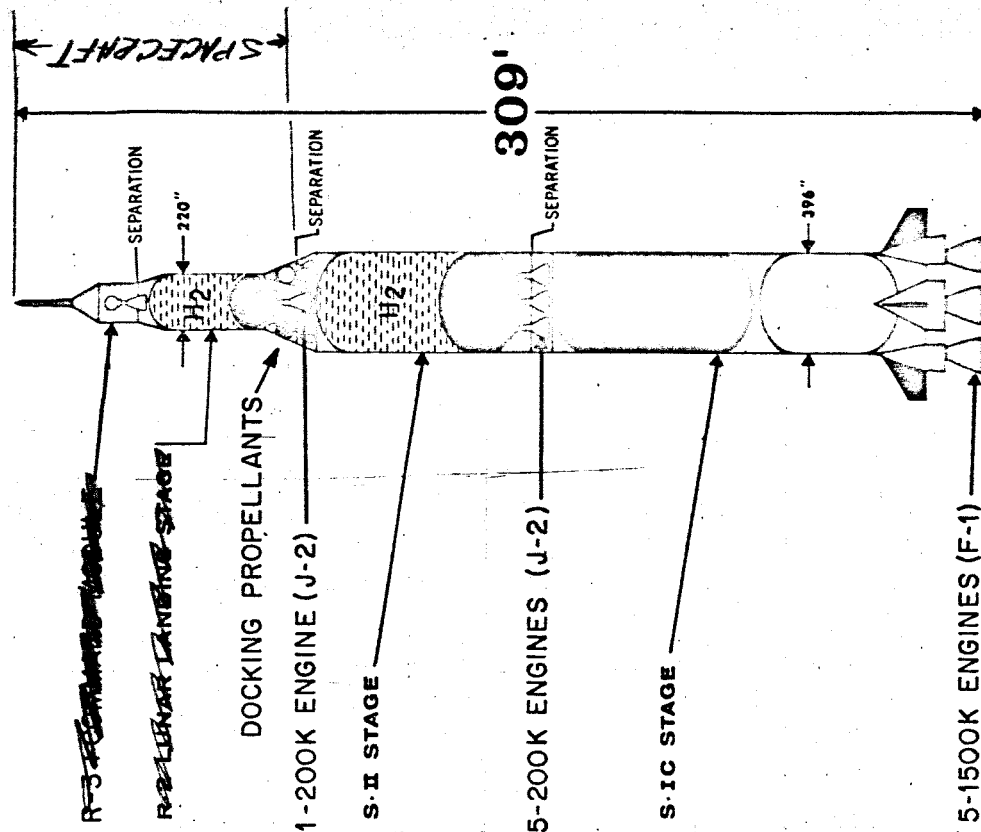


C-5 FOR ORBITAL OPERATIONS

CONNECTING MODE TYPICAL



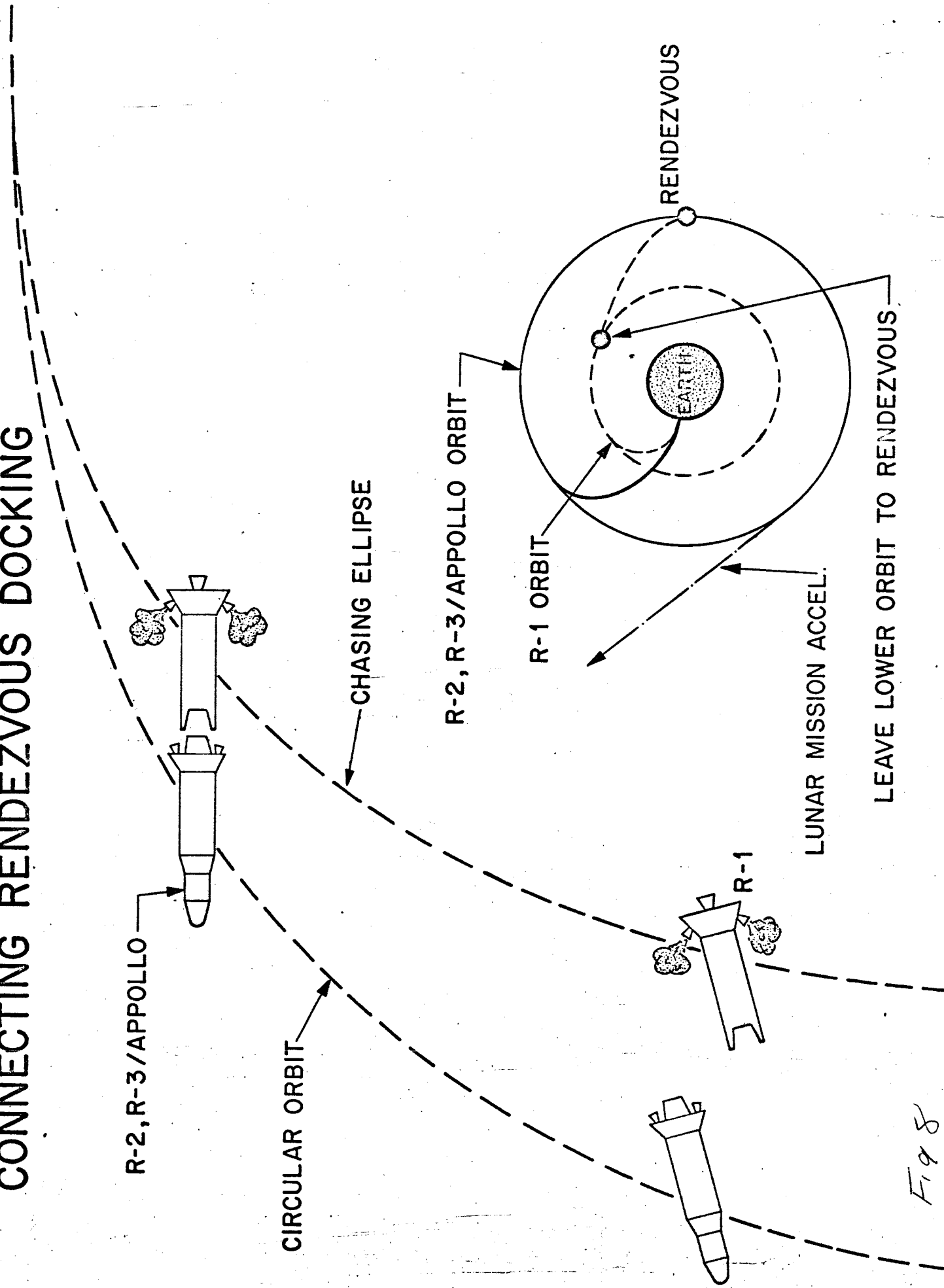
FIRST LAUNCH



SECOND LAUNCH

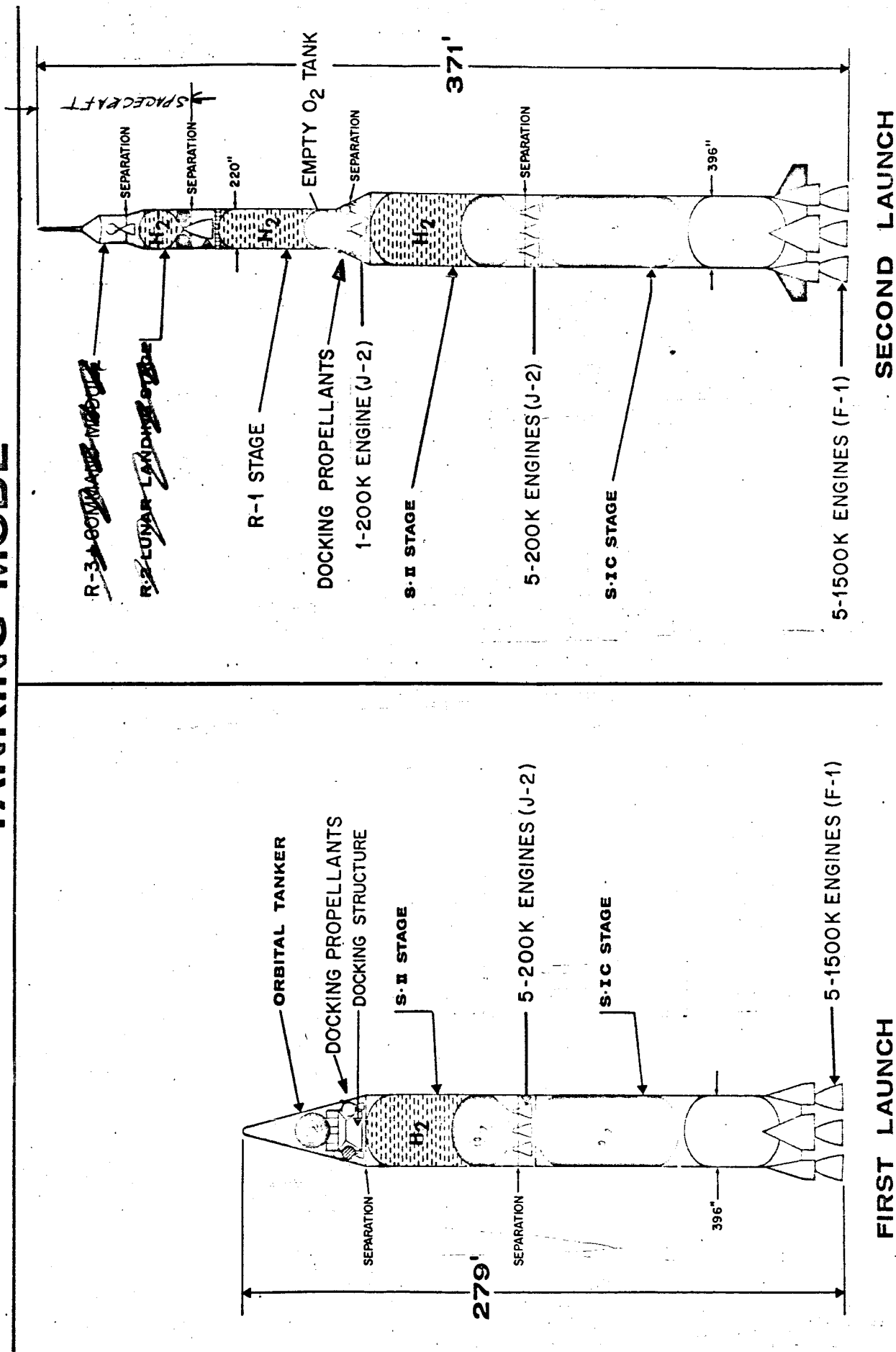
TYPICAL

CONNECTING RENDEZVOUS DOCKING



TYPICAL

C-5 FOR ORBITAL OPERATIONS TANKING MODE



TYPICAL

TANKER RENDEZVOUS DOCKING

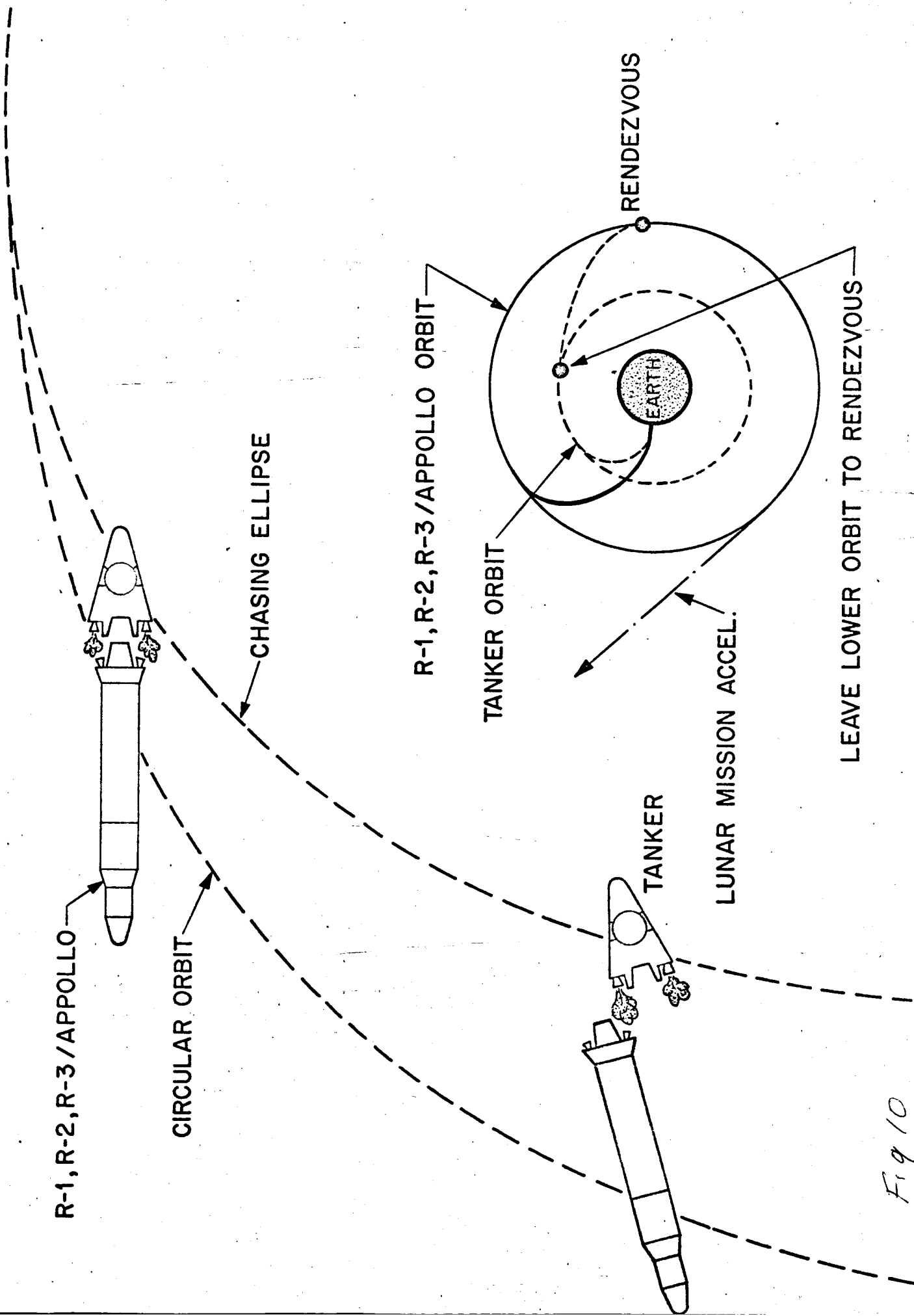


Fig 10

ENGINE DATA

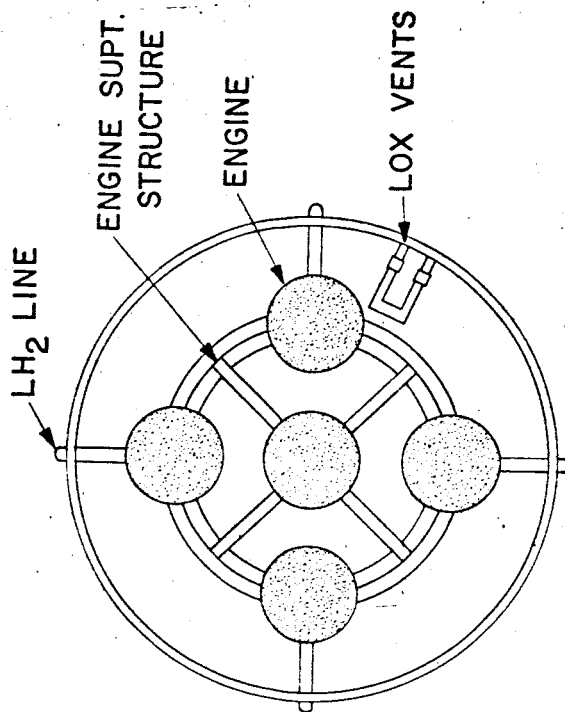
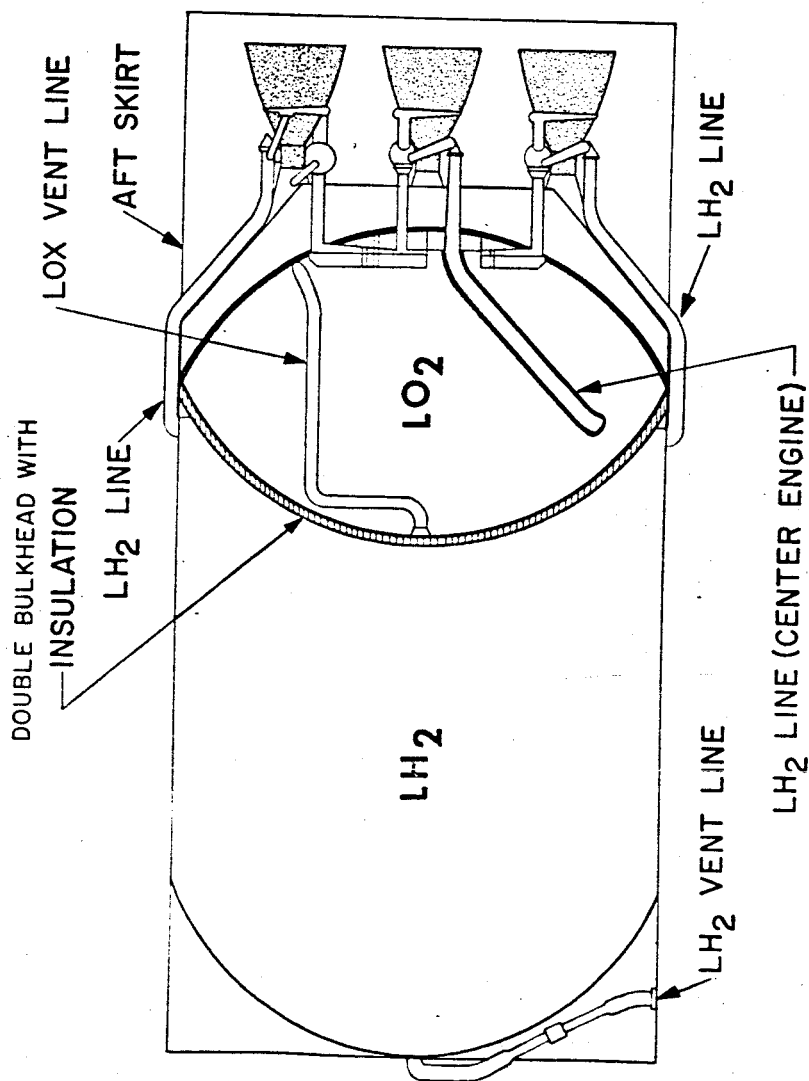
MANUFACTURER ~~ROCKETDYNE~~
 NPSH, OXIDIZER ~~65 FT.~~
 NPSH, FUEL ~~85 FT.~~
 GIMBAL RATE ~~10 DEGREES/SEC~~
 GIMBAL ACCELERATION ~~1 ROD/SEC.²~~

LEGEND

- 1 HEAT SHIELD
- 2 FOX PILL
- 3 FOX PILL
- 4 RETARD ROCKET
- 5 DEPRESSOR
- 6 FOX PILL
- 7 FOX PILL
- 8 FOX PILL
- 9 FOX PILL
- 10 FOX PILL
- 11 FOX PILL
- 12 FOX PILL

S-IC STAGE

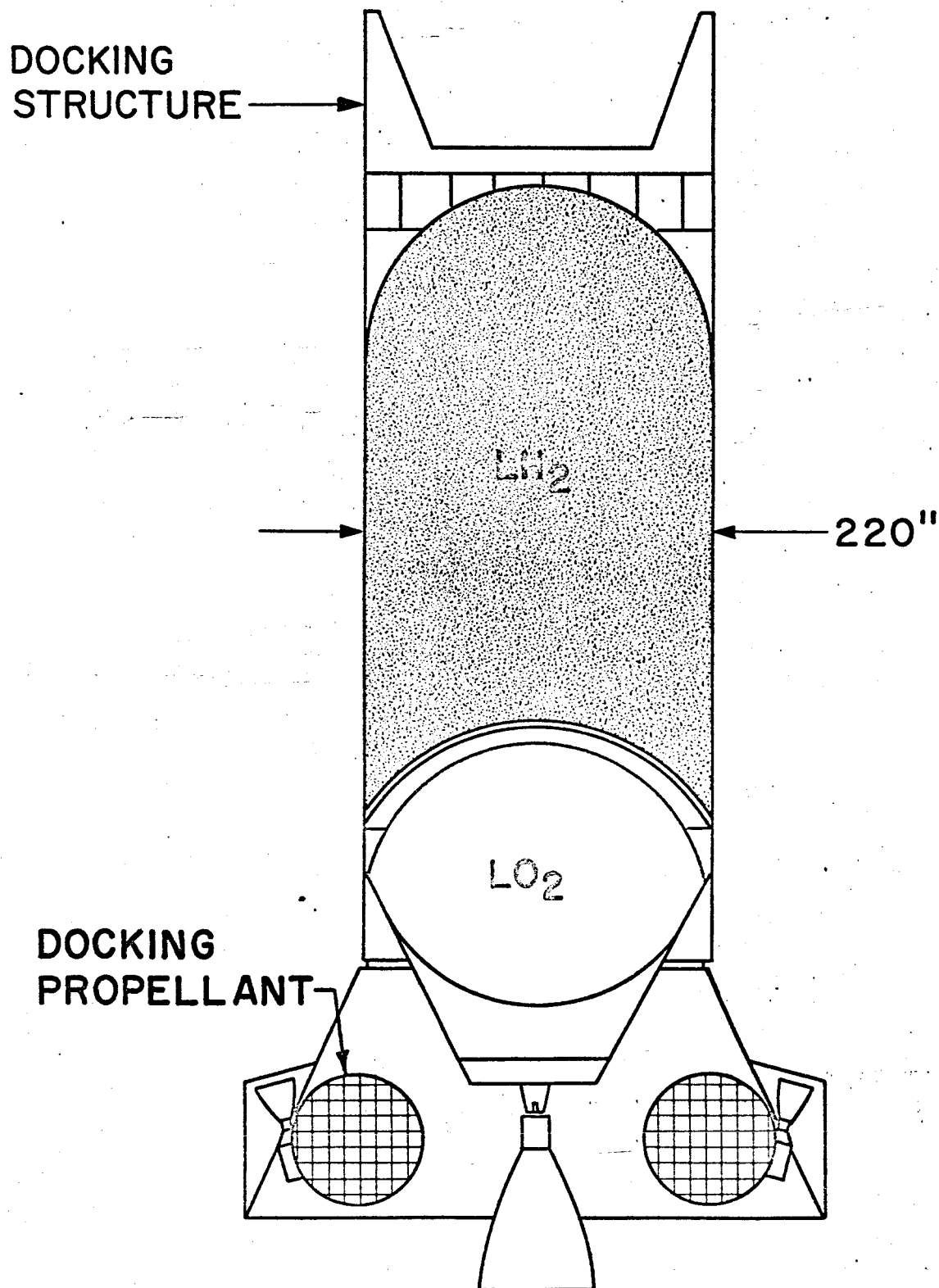
Note: This fig. to be replaced with a schematic type drawing similar to fig 12.



ENGINE DATA	
MANUFACTURER	ROCKETDYNE
NPSH, OXIDIZER	25 FT.
NPSH, FUEL	130 FT.
GIMBAL RATE	30 DEGREES/SEC.
GIMBAL ACCELERATION	2.25 ROD/SEC. ²

S-II STAGE
TYPICAL

TYPICAL
R-I



TYPICAL

C-5 FUELING MODE LOAD CURVES

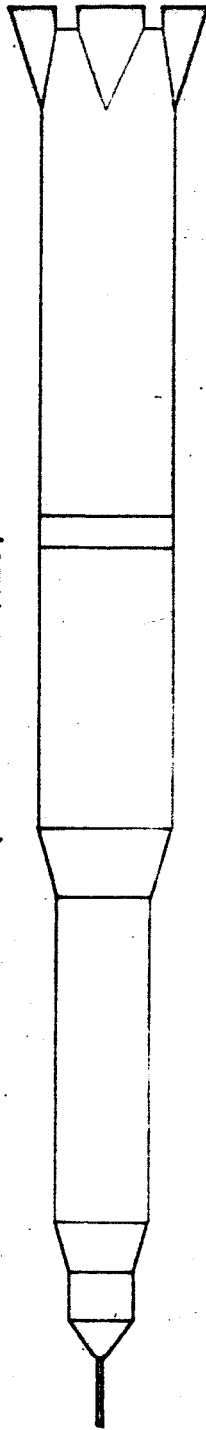
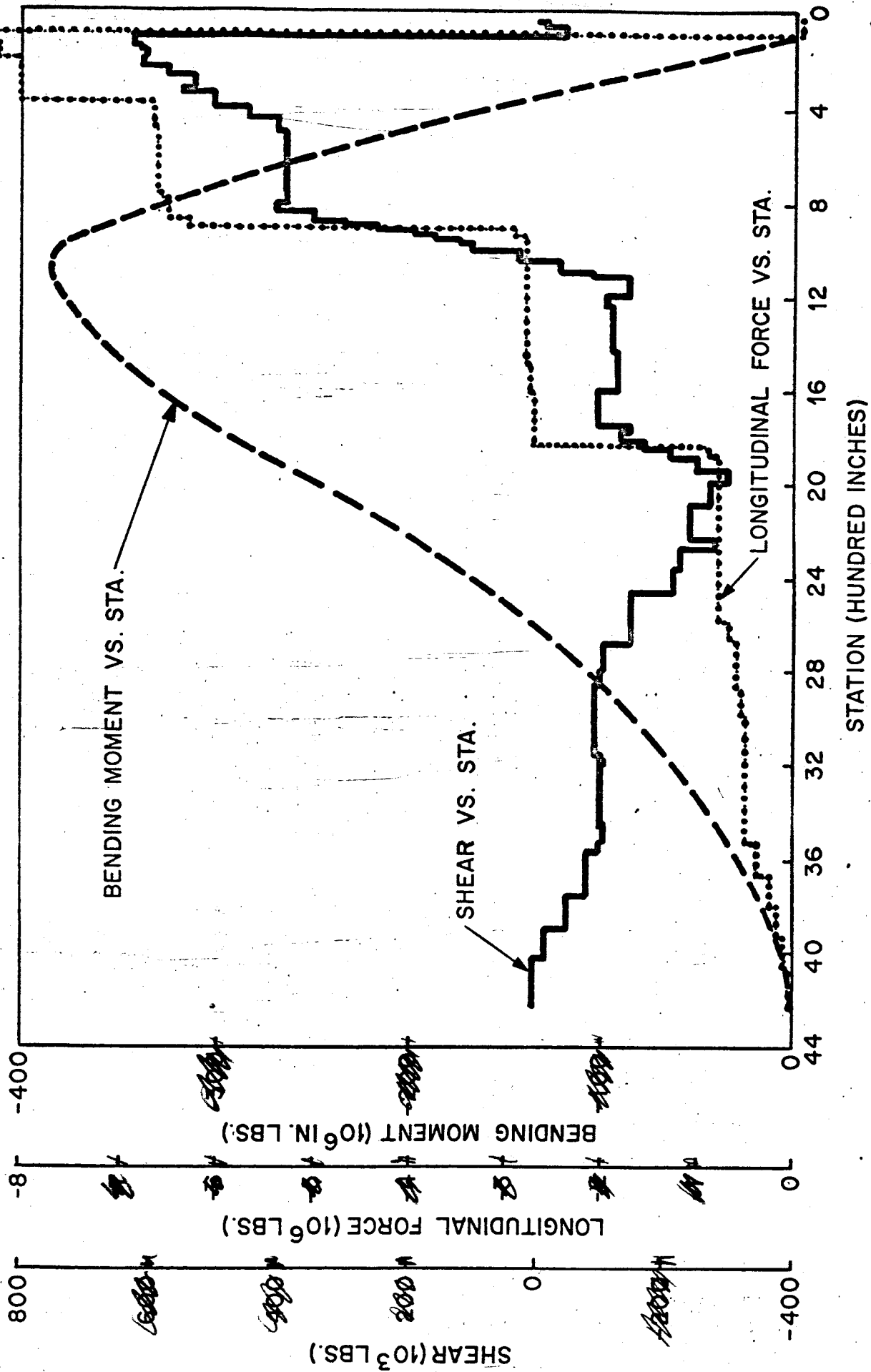
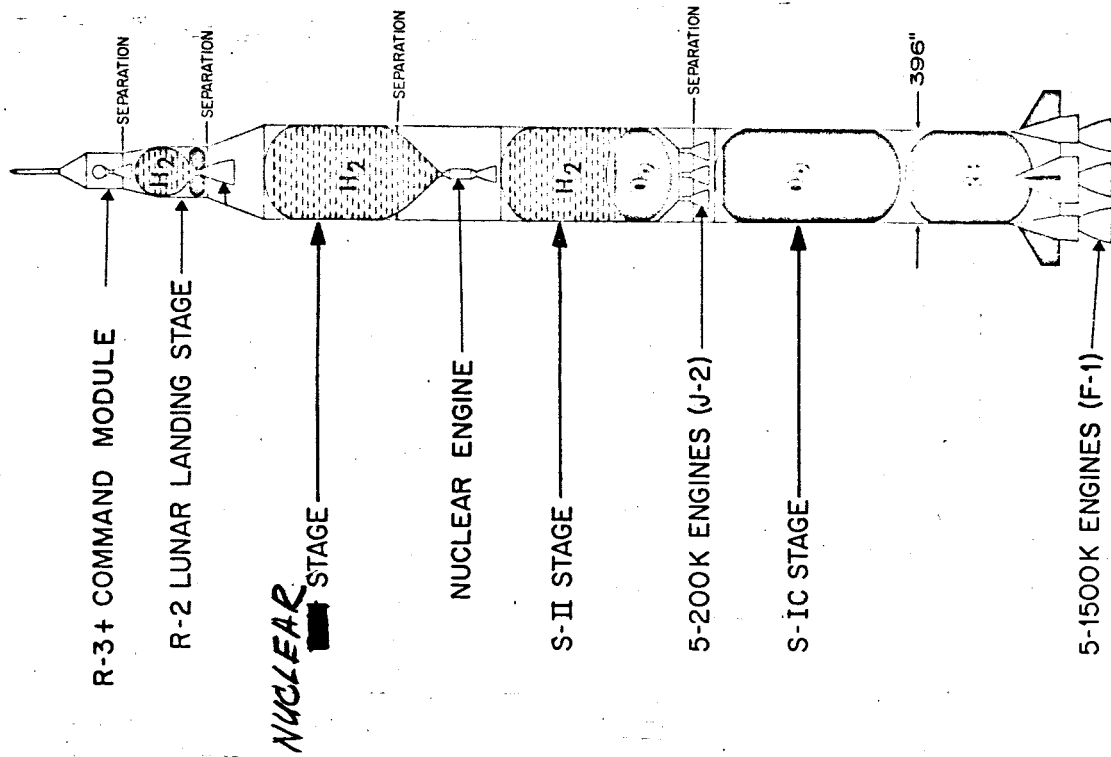


Fig 14

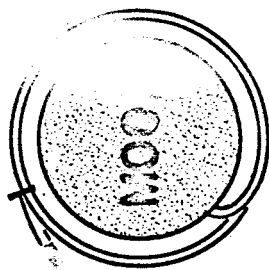
TYPICAL C-5 NUCLEAR



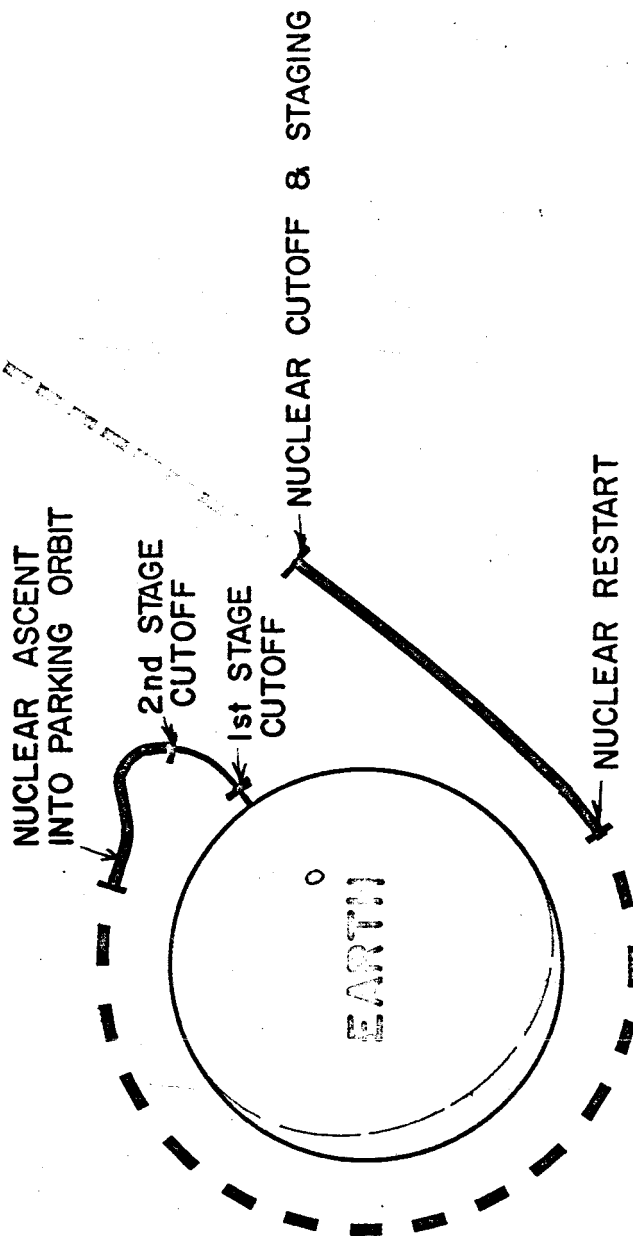
NUCLEAR MANNED LUNAR MISSION PROFILE

(TYPICAL)

BRAKING INTO LUNAR
ORBIT & DESCENT



CHEMICAL STAGE
STORABLE PROPELLANT
LANDING & RETURN



TYPICAL NOVA

